AC NO: 20-95

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ADVISORY CIRCULAR

DEPARTMENT OF TRANSPORTATION
FEDERAL AVIATION ADMINISTRATION

SUBJECT: FATIGUE EVALUATION OF ROTORCRAFT STRUCTURE

1. PURPOSE. This advisory circular sets forth acceptable means of compliance with the provisions of Federal Aviation Regulations, Sections 27.571 and 29.571 dealing with the fatigue evaluation of rotorcraft structure.

2. CANCELLATION. Appendix A, Main Rotor Service Life Determination, January 15, 1963, to CAM 6 is cancelled.

3. HOW TO GET THIS PUBLICATION.

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Initiated by: AFS-120

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SECTION 1. INTRODUCTION

1. BACKGROUND. The fatigue evaluation procedures outlined in this advisory circular are for guidance purposes only and are not mandatory nor regulatory in nature. Although a uniform approach to fatigue evaluation is desirable, it is recognized that in such a complex problem, new design features and methods of fabrication, new approaches to fatigue evaluation, and new configurations may require variations and deviations from the procedures described herein. Engineering judgment should therefore be exercised for each particular application.

   a. The flight structure of the rotorcraft is subject to vibratory stresses in practically every regime of flight. In addition, since it is a highly maneuverable aircraft that is capable of forward, rearward, sideward, vertical, and rotational flight, operating limitations due to fatigue are possible in practically all flight situations. For these reasons, it is required that special attention be focused on the fatigue evaluation of the flight structure of the rotorcraft.

   b. Fatigue evaluation of the flight structure is intended to verify structural reliability. Assurance of structural reliability starts with design, including choice of materials for resistance to crack initiation and/or propagation, detail design to minimize stress concentration, and specification of surface finishes, fits, etc. Design analysis should include estimation of expected flight loads, and estimation of resistance to fatigue. Fatigue strength should be based on past full scale fatigue tests and/or materials fatigue data with appropriate reductions for the variability in fatigue strength, size, shape, surface finish, and environments of the structure. In addition, design for fatigue should consider mode-of-failure analysis, areas susceptible to fatigue cracking, and methods to assure detectability of fatigue cracks. The residual strength of a cracked structure is an important consideration of fail-safe design.

   c. Assurance of structural reliability also includes manufacture and fabrication in accordance with design requirements and specifications, quality control to monitor compliance, and effective service inspection procedures.

d. Fatigue evaluation of the structure, measurement of flight loads and stresses, and evaluation of fatigue strength and/or fatigue crack propagation are the subjects of this advisory circular. There is some question whether a completely reliable method for the prediction of time to fatigue crack initiation and fracture exists. Nevertheless, one engineering approach to the subject is to use the "Linear Cumulative Damage Hypothesis." This hypothesis states that every cycle of stress above an "endurance limit" produces fatigue damage proportional to the ratio of cycles accumulated at the stress to fatigue "life" at that stress.

e. Laboratory tests of this hypothesis indicate that it is reasonably valid when the loading spectrum consists of stresses which are, in effect, random. Despite the lack of an adequate theory connecting this hypothesis with more basic properties of materials, it has been successfully used in a number of applications.

f. In addition, fatigue evaluation generally requires a method of accounting for the effect of steady loads and stresses on fatigue. Where the manufacturer does not provide other substantiating data, a Goodman diagram may be used to account for these effects.

g. In any rational fatigue evaluation, the following factors should be considered:

(1) Identification of the structure to be considered in the fatigue evaluation.

(2) The stresses associated with steady and maneuvering operating conditions expected in service.

(3) The frequency of occurrences associated with various flight conditions and the corresponding spectrum of loadings and stresses.

(4) The fatigue strength, fatigue crack propagation characteristics of the structure, and the residual strength of the cracked structure.

SECTION 2. FLIGHT STRAIN MEASUREMENT PROGRAM

2. GENERAL. Subsequent to design analysis, in which aircraft loads and associated stresses are derived, the stress level and/or loads are to be verified by a carefully controlled flight strain measurement program.

3. INSTRUMENTATION.

a. The instrumentation system used in the flight strain measurement program should accurately measure and record the critical strains under test conditions associated with normal operation and specific maneuvers. The location and distribution of the strain gages should be based on a rational evaluation of the critical stress areas. This may be accomplished by appropriate analytical means, supplemented, when deemed necessary, by strain sensitive coatings or photoelastic methods. The distribution and number of strain gages must define the load spectrum adequately for each part essential to the safe operation of the rotorcraft.

b. The corresponding flight parameters (airspeed, rotor r.p.m., center of gravity accelerations, etc.) should also be recorded simultaneously by appropriate methods. This is necessary in order to correlate the loads and stresses with the maneuver or operating condition at which they occurred.

c. The instrumentation system should be adequately calibrated and checked periodically throughout the flight strain measurement program in order to ensure consistent and accurate results.

4. PARTS TO BE STRAIN-GAGED. Fatigue critical portions of the rotor systems, control systems, fuselage, and supporting structure for rotors, transmission, and engine are to be strain-gaged. For rotorcraft of unusual or unique design, special consideration might be necessary to insure that all of the essential parts are evaluated.

5. FLIGHT REGIMES AND CONDITIONS TO BE INVESTIGATED.

a. Typical flight and ground conditions to be investigated in the flight strain measurement program are given in Table 1. Flight regimes that should be investigated for power-on and power-off operation are shown in Figures 1 and 2. For clarity, the parameters which define these regimes are included in these figures. As noted on Figure 1, complete coverage at 111 percent V_{NE} should be demonstrated for power-on operation. However, for power-off operation, Figure 2, complete coverage at 111 percent V_{NE} for maximum and minimum design r.p.m. need not be obtained if points are obtained at V_{NE} at both maximum and minimum design r.p.m. and at 111 percent V_{NE} at both maximum and minimum placarded r.p.m. as indicated in the figure.

b. The determination of flight conditions to be investigated in the flight strain measurement program should be based on the anticipated use of the helicopter, and, if available, on past service records for similar designs. In any event, the flight conditions considered appropriate for the design and application should be representative of the actual operation in accordance with the rotorcraft flight manual. In the case of multiengine helicopters, the flight conditions concerning partial engine-out operation should be considered in addition to complete power-off operation. The flight conditions to be investigated should be submitted in connection with the flight evaluation program. Suggested flight conditions for single-engine helicopters used in normal operation are shown in Table 1.

c. The severity of the maneuvers investigated during the flight strain survey should be such that it is unlikely that service use will be more severe.

d. All flight conditions considered appropriate for the particular design are to be investigated over the complete rotor speed, airspeed, center of gravity, altitude, and weight ranges in order to determine the most critical stress levels associated with each flight condition. In order to account for data scatter and to determine the stress levels present, a sufficient amount of data points should be obtained at each flight condition. In some instances, the critical weight, center of gravity, and altitude ranges for the various maneuvers can be based on past experience with similar designs. This procedure is acceptable where adequate flight tests are performed to substantiate such selections. The combinations of flight parameters that produce the most critical stress levels should be included in the fatigue evaluation.

SECTION 3. FREQUENCY OF LOADING

6. TYPE OF OPERATION. The probable types of operation (transport, utility, etc.) for the rotorcraft should be established. The type of operation can have a major influence on the loading environment. Normally, the rotorcraft should be substantiated for the most critical general type of operation with consideration of special occasional types of operation.

7. LOADING SPECTRUM. The spectrum allocating percentages of time or frequencies of occurrence to flight conditions or maneuvers is to be based on the expected usage of the rotorcraft. This spectrum is to be such that it is unlikely that actual usage will subject the structure to damage beyond that associated with the spectrum. Considerations to be included in developing this spectrum should include prior knowledge based on flight history recorder data, design limitations established in compliance with FAR's 27.309 or 29.309 and recommended operating conditions and limitations specified in the rotorcraft flight manual. The distribution of times at various forward flight speeds should reflect not only the relation of these speeds to V_{NE} but also the recommended operating conditions in the rotorcraft flight manual which govern V_C or cruise speed. Where possible, it is desirable to conduct the flight strain-gage program by simulating the usage as determined above, with continuous recording of stresses and loads, thus obtaining directly the stress/load spectra for structural elements. Table 1 contains typical percent of occurrences for the various flight conditions for single-engine piston helicopters used in utility operations. This table should be used only as a guide and should be modified as necessary for each particular rotorcraft.

SECTION 4. FATIGUE STRENGTH EVALUATION

8. GENERAL. Information to guide fatigue evaluation based on safe-life considerations leading to recommended replacement times is provided in this section. Although there is a large quantity of information available on the fatigue strength characteristics of material specimens, built-up specimens and parts, the prediction of strength of parts of new designs based on this information is less reliable than testing the actual part. Consequently, additional conservatism should be used with this method. However, in many cases the differences between past test specimens and the actual part (which involve such factors as stress concentration, size, and fretting) cannot be accounted for with a reasonable degree of accuracy. Therefore, it is usually necessary that the structural components be subjected to repeated load tests using information determined in the flight strain measurement program. Special operational or functional characteristics which could affect the fatigue strength should also be considered in the service life evaluation. Such factors as high blade operating temperatures due to tip jets or

turbine exhaust impingement on the tail rotor should be considered as well as other special operating conditions. In addition, effects of special purpose use such as hoist and sling operation, spraying, surveying, etc., should be considered if appropriate to the particular type. The fatigue strength should be evaluated by either of the methods outlined below, but full scale testing is considered more accurate.

9. <u>ANALYTICAL METHOD</u>. It is recognized that if allowable stress levels are established by acceptable means, and the stresses measured in flight are lower than these established levels, no fatigue testing is necessary.

a. <u>Simplified method</u>. The following techniques, based on the use of the Goodman diagram, are considered acceptable for establishing this allowable stress level:

(1) Estimate the mean endurance limit of the part from test results of specimens with similar stress concentrations. The test specimen material should be representative of the actual part and sufficient test data should be available to substantiate the mean endurance limit. The estimate should account for surface conditions, fabrication methods, fretting, size and shape effects, as well as differences in stress concentrations between the test specimen and the actual part. Referring to Figure 3, the mean endurance limit may be represented by a straight line drawn through the yield stress (point A on the horizontal axis) and the maximum oscillatory stress which the average specimen can withstand at a given steady stress (Point B) without failure for 10^7 to 5×10^7 cycles depending on the material.

(2) A factor of safety of 3 should then be applied to the mean endurance limit so that the slope of line AC would be 1/3 of line AB. A smaller factor is acceptable when substantiated by a sufficient number of tests on similar parts in similar applications.

(3) If the flight strain measurements indicate that all of operating stresses fall below the operating boundary line (AC), no fatigue testing is necessary. When the measured stresses are above the operating boundary line, however, fatigue testing of the actual parts should be conducted.

(4) Caution should be exercised in the application of the analytical method above, particularly when the following items are involved:

(a) Large parts in proportion to the laboratory specimens.

(b) Irregularly shaped parts containing numerous or superimposed fillets, holes, threads, or lugs.

(c) Parts of unique design for which no past service experience is available.

(d) Parts subject to fretting.

(e) Bolted or pinned connections.

b. Rational methods. Methods may be used which do not involve full scale testing but which apply the variables of fatigue strength with a calculation of retirement times in a manner that provides equivalent reliability to the fatigue testing and simplified methods and is acceptable to the Administrator.

10. TESTING METHODS. The fatigue strength of the flight structure may be determined in appropriate laboratory tests and evaluated in terms of a loading spectrum. The strength indicated by the test results should be reduced by a factor such that a replacement time based directly on this reduced strength level and the loading spectrum of paragraph 7 will assure that the probability of failure is extremely remote. This reduction factor should be based on consideration of the number of specimens tested, the variability of the fatigue results, the effects of service use, and, where available, previous test data for the same material or similar components as well as service experience.

The test methods outlined below are considered acceptable.

a. S-N Curves.

(1) Fatigue tests should be conducted over a range of oscillatory stresses or loads to define the S-N curves. Fatigue tests should be performed at steady stresses or loads representative of those occurring in flight.

(2) In order to determine the mean fatigue strength and the variability in fatigue strength, it is necessary to test a number of specimens in establishing S-N curves. In order to account for the variability in fatigue strength, a reduction factor should be applied to the mean curve in arriving at a working S-N curve. This factor should include consideration of the number of specimens tested,

the variability of the fatigue results, and, where available, previous test data on the same material or similar components, as well as service experience. Where new materials or designs are being evaluated, it is recommended that a larger reduction factor be used until such time as additional test data justifying a change are available. The mean and reduced S-N curve should reflect the curve shapes of typical published S-N data on notched or unnotched specimens, as applicable. The reduced S-N curve and the loading spectrum of paragraph 7 should be used in determining replacement times. Figure 4 represents the method of constructing a typical S-N curve from the fatigue test data.

b. Spectrum tests. The establishment of replacement times based on fatigue tests in which each specimen is subjected to a spectrum of loading is to include the following considerations:

(1) Definition of the test loading spectra based on either:

(a) Analysis, supported by extrapolation of available load history data or prior knowledge where available, or

(b) Stress histories based on flight test data obtained for flight and ground conditions and maneuvers considered appropriate for the particular rotorcraft, and a spectrum allocating percentages of time or frequencies of occurrence to these flight and ground conditions and maneuvers.

(2) Fatigue tests in which the loading spectra are applied such that effective randomization of loadings is obtained.

(3) Unless performed prior to step (1), determining by flight test the stress levels associated with each flight condition and maneuver considered appropriate for the particular rotorcraft.

(4) Assignment of replacement times. The fatigue test results should be evaluated in terms of the loading spectrum of paragraph 7 (if different than test spectrum) and reduced

by a factor considering the number of specimens, the fatigue strength variability, and applicable prior data, in arriving at a replacement time.

c. Major system tests. Another method of determining the replacement times is to perform a fatigue test or tests of the major systems. Examples of such testing are whirl tests, tiedown tests, and bench tests. The test results should be evaluated in terms of the loading spectrum of paragraph 7 (if different from test loading) and reduced by a factor considering number of specimens tested, variability in results, and applicable prior test data in arriving at a replacement time.

11. COMBINATION OF REPLACEMENT TIME AND FAIL-SAFE EVALUATION. It may be possible to extend the replacement time of safe-life components which exhibit limited fail-safe capability by using a combination of the safe-life and fail-safe concepts. This is accomplished by evaluating both the fatigue strength and fail-safe characteristics as described elsewhere in this circular and by assigning both a replacement time and inspection period to these components. The replacement time may then be based on the combined probability of not initiating a fatigue crack at or before the replacement time and the probability that the crack if initiated will be detected prior to catastrophic failure or loss of limit load (or maximum attainable load, whichever is less) carrying capability. The probability of detection should be based on consideration of the inspection effectiveness, the inspection interval, and the fatigue life remaining after an obvious partial failure. A lower strength reduction factor, commensurate with this probability of detection, may then be used in the determination of the replacement time.

12. EXTENSION OF REPLACEMENT TIME. Parts should be replaced or retired at the established service period unless additional data indicate that an extension of the service period is justified. Important factors in the consideration of such extension would be:

a. Recorded load data. Recording load data entails instrumenting aircraft in service to obtain a representative sampling of actual loads experienced. The data measured should include airspeed, altitude, and rotor speed versus time, or the airspeed, altitude, and strain ranges versus time, or similar data. The data obtained by instrumenting aircraft in service should provide a basis for correlating the estimated loading spectrum with the actual service experience.

b. Additional analyses and tests. If test data and analyses based on repeated load tests of additional specimens are obtained, a reevaluation of the initial strength reduction or scatter factor may be made.

c. Tests of parts removed from service. Where conservatism was used in initial calculation of replacement times because of lack of knowledge of service environment, repeated load tests of replaced parts may be utilized to reevaluate the initial scatter factor selected. The tests should closely simulate service loading conditions.

d. Rework of the structure. In some cases, rework of the structure may result in an increase in replacement time.

SECTION 5. FAIL-SAFE STRENGTH EVALUATION

13. GENERAL. The fail-safe strength evaluation of the flight structure is intended to insure that, should fatigue cracks initiate, the remaining structure will withstand service loads without failure until the cracks are detected. The fail-safe evaluation generally encompasses establishing the components which are fail-safe, defining the loading conditions and extent of damage for which the structure is to be designed, conducting structural tests and analysis to substantiate that the design objective has been achieved, and establishing inspection programs to assure detection of fatigue damage. On components predominantly loaded by centrifugal force, care should be taken in selecting limit load to assure that it is the maximum expected in service. Design features which may be used in attaining a fail-safe structure are:

a. Selection of materials and stress levels that provide a controlled slow rate of crack propagation combined with high residual strength after initiation of cracks.

b. Design to permit detection of cracks including the use of crack detection systems, in all critical structural elements before the cracks can become dangerous or result in appreciable strength loss, and to permit replacement or repair.

c. Use of multipath construction and the provision of crack stoppers to limit the growth of cracks.

d. Use of composite duplicate structures so that a fatigue crack or failure occurring in one element of the composite member will be confined to that element and the remaining structure will still possess appreciable load-carrying ability.

e. Use of backup structure wherein one member carries all the load, with a second member available and capable of assuming the extra load if the primary member fails.

14. IDENTIFICATION OF CRITICAL PORTIONS OF FLIGHT STRUCTURE. Those portions of the flight structure which may be critical in fatigue should be identified. Typical portions of the structure are:

a. Rotor blades and attachment fittings.

b. Rotor heads, including hubs, hinges, dampers.

c. Control system components, including control rods, servos, swashplates.

d. Rotor supporting structure.

e. Fuselage, including stabilizers and auxiliary lifting surfaces.

15. EXTENT OF FAIL-SAFE DAMAGE. The extent of the partial failure is to be such that it would be readily detectable during the specified inspection. It may involve complete failure of a principal element, failure of more than one element, or only a partial failure of an element depending on the rate of crack propagation, the ease of detection, and the inspection interval. Damage in inaccessible areas should extend into inspectable areas.

Typical examples of the fatigue damage which should be considered are outlined below:

a. Cracks emanating from the edge of structural openings or cutouts which can be readily detected by visual inspection of the area.

b. A circumferential or longitudinal skin crack in the basic fuselage structure of such a length that it can be readily detected by a visual inspection of the surface area.

c. Complete severance of interior frame elements or stiffeners in addition to a visually detectable crack in the adjacent skin.

d. Failure of one element where dual construction is utilized in components.

e. Failure of primary attachments, including control hinges and fittings.

16. DETERMINATION OF PROBABLE CRACK LOCATIONS. The probable crack locations are to be determined by tests, analysis, or both. In cases of unusually critical or complex components or when initial fatigue loadings may affect the rate or mode of cracking, the probable crack locations should be determined by fatigue test. When determination is made by analysis, sound engineering judgment should be used and a variety of factors such as the following taken into account:

a. Conducting an analysis to locate areas of maximum stress and low margin of safety.

b. Conducting strain surveys on undamaged structure to establish points of high stress concentration as well as the magnitude of such concentration.

c. Examining static test results to determine locations where excessive deformation occurred.

d. Determining from fatigue analysis where cracks may initiate.

e. Selecting locations in an element where the stresses in adjacent elements would be the maximum with that element failed.

f. Selecting partial fracture locations in an element wherein high stress concentrations are present in the residual structure.

g. Assessing design detail areas which are prone to fatigue damage based on service experience records of similarly designed components.

17. FAIL-SAFE DEMONSTRATION. It is to be demonstrated by analysis, tests, or both, that the structure with the partial failures as defined in paragraphs 15 and 16 can withstand the maximum load and the repeated loads expected in service during the period prior to detection. The repeated loads should be as defined in the loading spectrum of paragraph 7 and the structure should be capable of supporting this loading after a partial failure for a sufficient time with respect to the inspection interval to assure that catastrophic

failure is extremely remote. The loading spectrum should include at least one application of limit load. In test demonstrations, the damage may be initiated or simulated by cuts made with a fine saw, sharp blade, or guillotine in those cases where it is not necessary and not practical to produce fatigue cracks by tests. In those cases where damage is simulated at joints or fittings, bolts may be removed to simulate failure if this condition would be representative of an actual failure. In some instances, the fail-safe characteristics may be shown analytically. The analytical approach may be used when the structural configuration involved is essentially similar to one already verified by fail-safe tests, whether on a previously approved type design, or on other similar areas of the design currently being evaluated. The analytical approach may also be used when: (1) it can be shown that the failure would be detected considerably before the critical crack length is approached; (2) the margins of safety resulting from the analysis are well in excess of the fail-safe residual static strength level; and (3) the stress levels in the partially failed structure and the design are such as to assure adequate crack propagation time relative to the inspection interval.

18. INSPECTION. Detection of fatigue cracks before they become dangerous is the ultimate control in insuring the fail-safe characteristics of the structure. Therefore, the manufacturer should provide sufficient guidance information to assist operators in establishing the frequency and extent of the repeated inspections of the critical structure.

TABLE 1

Percent occurrence

I.	GROUND CONDITIONS			1.5
	(a)	Rapid increase of r.p.m. on ground to quickly engage clutch		0.5
	(b)	Taxiing with full cyclic control		.5
	(c)	Jump takeoff	(1)	
	(d)	Ground-air-ground cycle	(1)	
II.	HOVERING			2.0
	(a)	Steady hovering		.5
	(b)	Lateral reversal		.5
	(c)	Longitudinal reversal		.5
	(d)	Rudder reversal		.5
III.	FORWARD FLIGHT-POWER ON			87.5
	(a)	Level flight - 20% V_{NE}		1.0
	(b)	Level flight - 40% V_{NE}		3.0
	(c)	Level flight - 60% V_{NE}		18.0
	(d)	Level flight - 80% V_{NE}		25.0
	(e)	Maximum level flight (but not greater than V_{NE})		15.0
	(f)	V_{NE}		3.0
	(g)	111% V_{NE}		.5
	(h)	Right turns - 30, 60, 90% V_{NE}	(2)	3.0
	(i)	Left turns - 30, 60, 90% V_{NE}	(2)	3.0
	(j)	Climb (Takeoff power)		2.0
	(k)	Climb (Max. continuous power)		4.0
	(l)	Change to autorotation from power-on-flight - 30, 60, 90% V_{NE}		1.5
	(m)	Partial power descent (including condition of zero flow through rotor)		2.0
	(n)	Cyclic and collective pull-up from level flight	(2)	
	(o)	Pushovers	(2)	
	(p)	Gusts	(2)	
	(q)	Quick stops	(2)	
	(r)	Flares	(2)	
	(s)	Lateral reversals at V_H		.5
	(t)	Longitudinal reversals at V_H		.5
	(u)	Rudder reversals at V_H		.5
	(v)	Landing approach		3.0
	(w)	Sideward flight		.5
	(x)	Rearward flight		.5
IV.	AUTOROTATION - POWER OFF			9.0
	(a)	Steady forward flight		2.0
	(b)	Rapid power recovery from autorotational flight		.5
	(c)	Right turns - 30, 60, 90% V_{NE}	(2)	1.0
	(d)	Left turns - 30, 60, 90% V_{NE}	(2)	1.0

(e)	Lateral reversals--	.5
(f)	Longitudinal reversals-----------------------------------	.5
(g)	Rudder reversals---	.5
(h)	Cyclic and collective pull-ups-------------------------(2)	
(i)	Landing approach---	2.0
(j)	Flares---(2)	
		100.0

(1) One flight every 10 minutes with less frequent rotor stops.

(2) A vertical load factor frequency curve should be developed that is representative of the more critical types of operation. The time spent in each turn should be adjusted to give the specified per cent of occurrence.

NOTE: DASHED LINES IN FIGURES 1 & 2 INDICATE TEST BOUNDARIES. CROSS-HATCHED AREAS INDICATE OPERATING REGIMES.

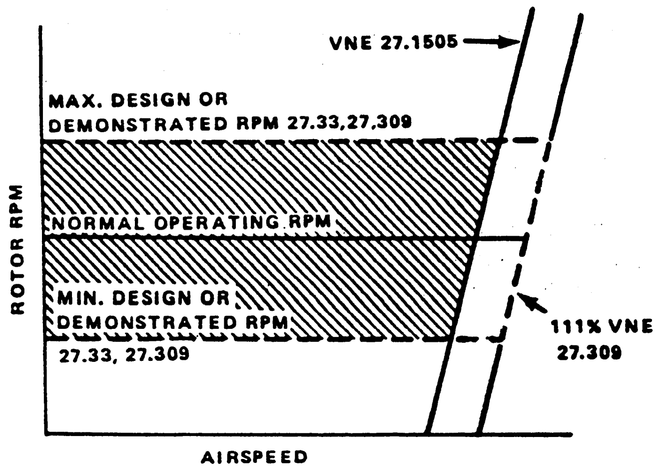

FIGURE 1. POWER ON–
ROTOR R.P.M. AIRSPEED ENVELOPE

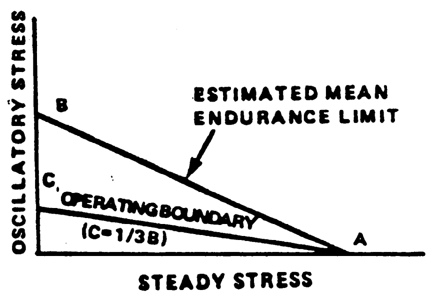

FIGURE 3. Oscillatory Stress vs. Steady Stress

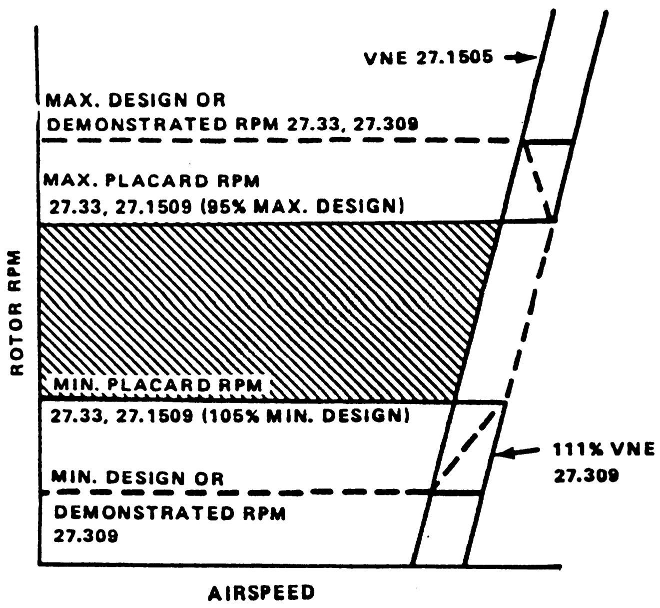

FIGURE 2. POWER OFF–
ROTOR R.P.M. AIRSPEED ENVELOPE

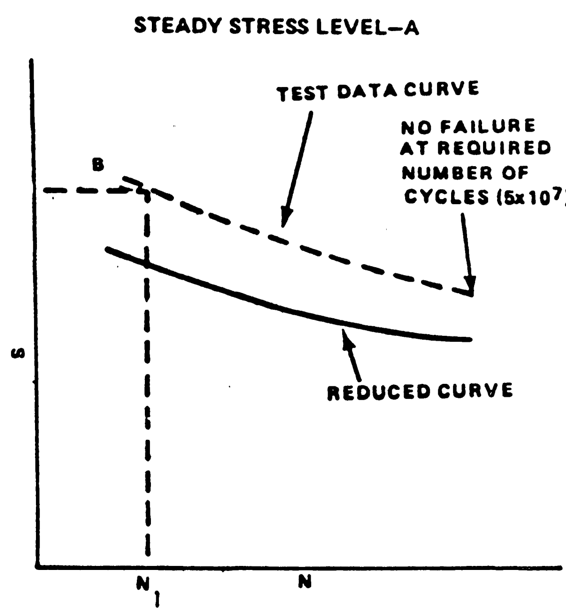

FIGURE 4. Stress vs. Number of Cycles